Low Thrust Mission Trajectories to Near Earth Asteroids

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The discovery of 2016 HO3 and its classification as a quasi-satellite has sparked a stronger interest towards Near Earth Asteroids (NEAs). This work presents low-thrust low-power mission designs to various NEAs using an EELV Secondary Payload Adapter (ESPA). A global trajectory optimizer (EMTG) was used to generate mission solutions to a select 13 NEAs using a 200 watt BHT-200 thruster as a proof of concept. The missions presented here demonstrate that a low-cost electric propulsion ESPA mission to NEAs is a feasible concept for many asteroids.

I. Introduction

The official NASA plan for the exploration of Mars was announced in 2015, using a flexible path to get there. This flexible path promoted the concept of exploitation of the Near Earth Asteroids (NEAs), including the challenging mission of bringing an asteroid into cis-lunar space. Many manned-target missions require a surveying mission, similar to the Lunar Reconnaissance Orbiter.[1] The Smallsat paradigm promotes many smaller and cheaper spacecrafts that could exploit these missions. A cheap smallsat, such as the spacecraft proposed here with a low power electric propulsion system, could easily explore the NEAs relatively cheaply.

The primary goal of this work was to generate mission designs to various NEAs as a proof of concept. Constraints on thrust/power and overall payload mass were imposed to simulate the characteristics of a candidate small spacecraft design. The following sub-sections will discuss how the NEAs were selected, the constraints on the mission design, and the global optimizer used to generate the solutions.

II. Near Earth Asteroids

According to the Center for Near Earth Object Studies (CNEOS) at the Jet Propulsion Laboratory (JPL), there are over 2000 NEAs for which information on the physical characteristics as well as ballistic trajectory information are available.[2] Furthermore, CENOS has generated contour plots which describe the total impulsive mission delta-V as a function of departure date and mission duration for all of these NEAs. These plots help determine the best launch dates in terms of mission resources required and helped determine the low-thrust mission parameters for each NEA as will be discussed in further sections.

In order to generate a much smaller list of NEAs for this study, the following constraints were placed on the list of NEAs archived in CNEOS: a total ΔV of less than 6 $\frac{km}{s}$, total duration of less than 360 days, a stay greater than 8 days, and a launch window from 2020 to 2035. The constraints were based off the proposed Asteroid Redirect Mission (ARM) to ensure realistic parameters for mission design.[3] It was scheduled for launch in 2021 with potential NEA targets. Two of the targets (Bennu and 2008 EV5) fall relatively close to the mission constraints listed above. The constraints for this study generally focused on asteroids closer than the ones considered for the ARM mission due to propellant and thruster restrictions (mentioned below).

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From the list of NEAs resulting from these constraints, preliminary mission tests were run in EMTG to see which of these are feasible targets based on the limited propulsion available. A final list consisting of 13 NEAs was generated for the purpose of this study. Table 1 lists these NEAs, their physical parameters, and optimal launch dates ranges. It lists the 13 NEAs used in the study along with optimal launch dates, minimum delta-V and minimum duration solutions. These targets were acquired from CNEOS, where optimal launch dates (within the 2020-2035 launch period discussed below) were approximations taken from their trajectory plots. Minimum delta-V and minimum duration solutions are from ballistic approximations shown on CNEOS. All of the trajectory plots are obtained from CNEOS and are shown in Figure 1. The color bar represents the delta-v required in $\frac{km}{s}$. 2016 WF9 is not shown since there were no optimal launch dates found (left blank in Table 1).

NEA	Optimal Launch Dates	Estimated Diameter (m)	Minimum Delta-V (km/s)	Minimum Time (days)
2016 TB18	01-01-2024 to 01-01-2027	19-85	5.55	114
2015 VO105	01-01-2026 to 01-01-2033	26-118	5.67	98
2015 JD3	01-01-2025 to 01-01-2029	13-59	4.84	66
2014 KF39	01-01-2029 to 01-01-2034	15-68	5.41	338
2013 WA44	01-01-2026 to 01-01-2030	32-142	5.96	160
2012 UV136	01-01-2020 to 01-01-2030	14-62	5.05	282
2012 PB20	01-01-2023 to 01-01-2034	18-81	5.28	346
2012 BB14	01-01-2022 to 01-01-2025	17-78	5.18	306
2011 BP40	01-01-2020 to 01-01-2302	14-65	5.84	82
2009 HC	01-01-2023 to 01-01-2026	20-89	4.43	66
2000 SG344	01-01-2024 to 01-01-2035	20-89	3.55	34
1999 CG9	01-01-2032 to 01-01-2035	16-71	5.25	66
2016 WF9		6.9-31	11.86	362

Table 1: List of all NEAs selected for this study along with some mission parameters.

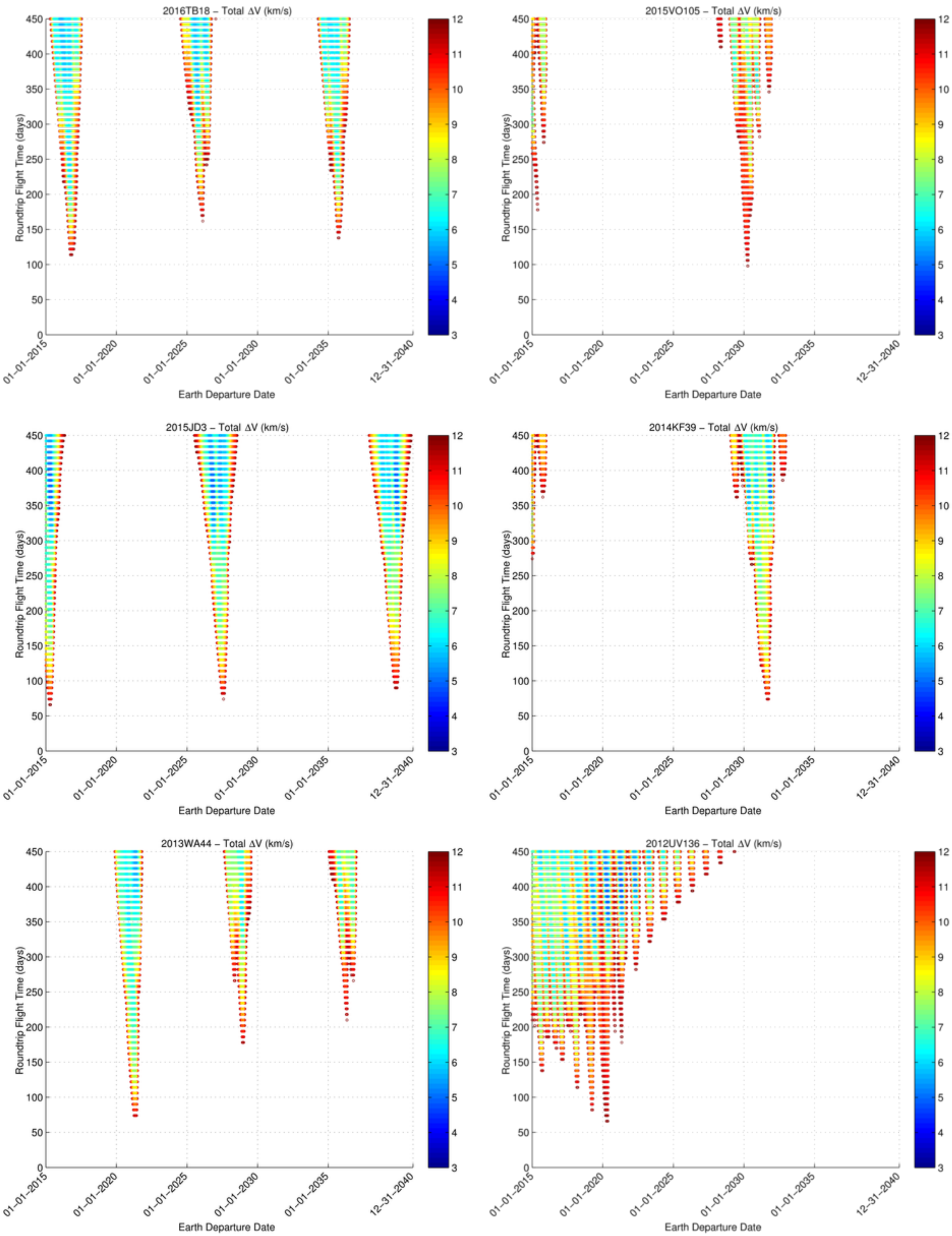

Figure 1: Trajectory plots for NEAs from CNEOS. The color bar represents the delta-v required in $\frac{km}{s}$ Courtesy NASA/JPL-Caltech

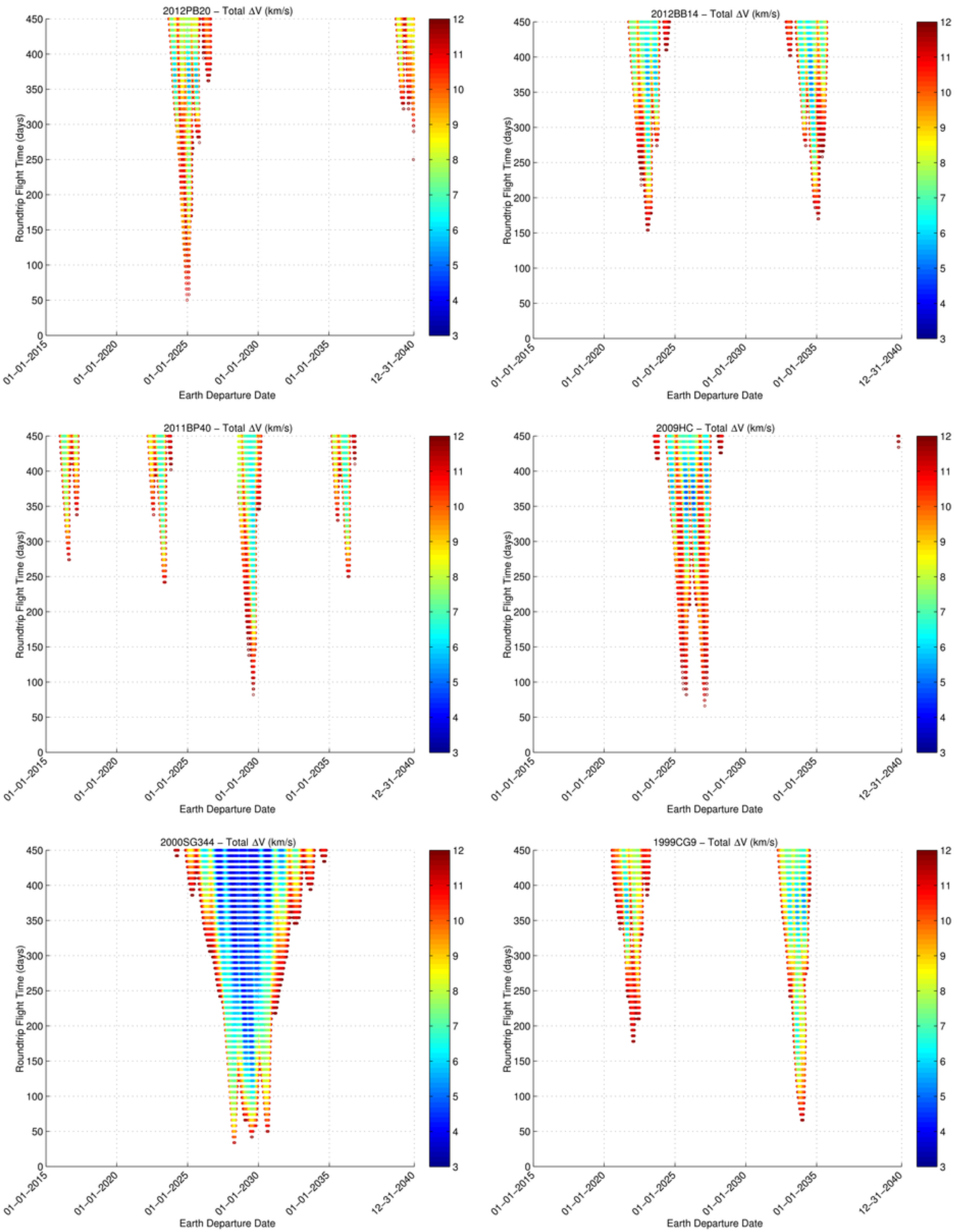

Figure 1: Trajectory plots for NEAs from CNEOS. The color bar represents the delta-v required in $\frac{km}{s}$. Courtesy NASA/JPL-Caltech

III. EMTG/Mission Constraints

The Evolutionary Mission Trajectory Generator (EMTG) is a GSFC in-house fully-validated NASA flight dynamics low thrust global optimizer that provides solutions for various mission criteria.[4] The mission constraints discussed above were entered into ETMG for each NEA. All of the results were optimized for maximum final mass in order to minimize the amount of propellant used. The following subsections explain each of the constraints placed upon the designed missions.

A. Launch/Departure

Missions were designed to launch from an Evolved Expendable Launch Vehicle (EELV) Secondary Payload Adapter (ESPA) attached to a launch vehicle of a primary mission. In the coming years, a multitude of lunar missions are set to occur, beginning with the EM-1 ICPS nominally scheduled for 2018. Therefore, an overall launch window was defined between 2020-2035. The actual launch date is NEA specific and is located within a small window (shown in table 1) based on trajectory plots provided by CENOS. For 2016 WF9, the default 2020-2035 launch window was used.

Each mission in EMTG was set to a direct launch from a circular orbit derived from the EM-1 ICPS launch vector due to limitations of launching from a elliptical parking orbit in EMTG. An approximation was assumed that the orbit right after launch for any lunar mission will be a circular orbit at 20,000 km which is the semi-major axis of the EM-1 orbit after launch. It is also assumed that any upcoming lunar mission will result in this circular orbit and therefore is used as a departure orbit for all NEAs missions designed in this study.

The EM-1 ICPS launch vector represents an eccentric orbit ($e = 0.96$). In order to derive a better understanding of the effects of the above approximation on mission design, a simple test case was designed. Assuming the worst case scenario is at apogee (lowest velocity) with a semi-major axis equal to the circular orbit stated above, the energy required to transfer to the circular orbit was calculated. Using the BHT-200 Busek Low Power Hall Thruster (discussed below) , it would take approximately 22 days to change orbits, with this being the upper limit on time required and spends about 2 kg of propellant.

B. Spacecraft Design

The spacecraft concept for this study is an ESPA-class smallsat. A candidate bus was identified to perform mission design.

Ball Aerospace has investigated modifying one of their standard smallsat busses with an electric propulsion system, specifically for a BCP-100 bus. Their bus provides all of the subsystems needed, as well as leaving an additional 46.9 kg available for propellant and payload. A figure of the proposed bus is shown in Figure 2 on a work stand. The bus provides all of the avionics for the payload, and 200W of power (solar arrays not shown).

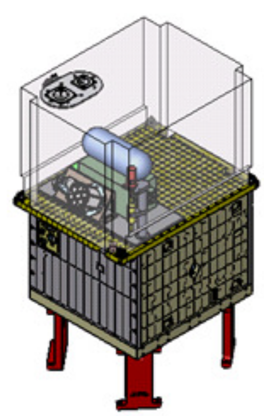

Figure 2: The BCP-100 bus modified with a small electric propulsion system. A larger tank was used for this study.

The PSI 80386-101 (4.6 kg dry mass, 2500 psi MEOP) is an optional propellant tank for the BCP-100 bus that allows for sufficient propellant to be contained for the mission and still fit inside the spacecraft envelope. This tank has more than sufficient capacity for the mission, as shown in Figure 3. This leaves 40 kg for propellant, and 2.3 kg for an instrument payload.

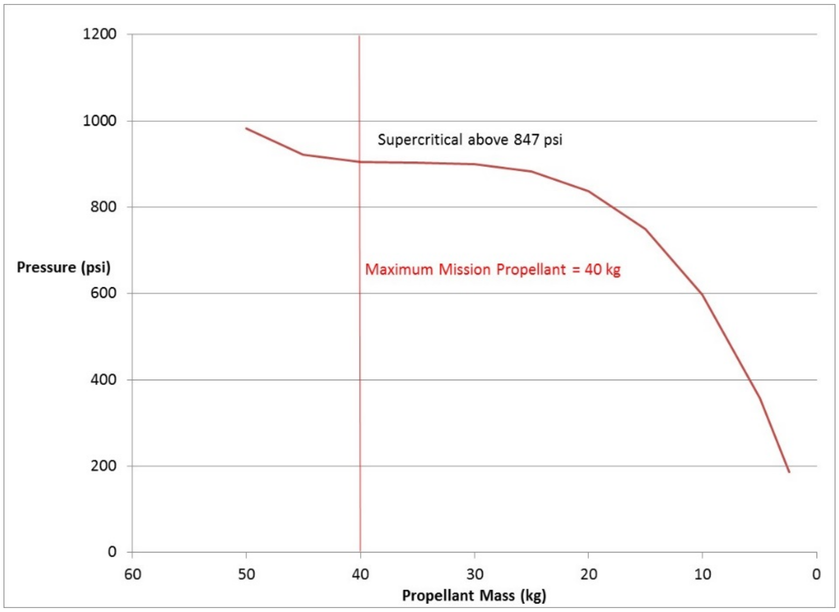

Figure 3: The sizing of a larger notional propellant tank.

In order to properly simulate a low thrust mission, the BHT-200 Busek Low Power Hall Thruster was used for each of the missions. According to its specifications,[5] it is capable of operating in the 100-300 W range and can produce a nominal 13 mN thrust (at 200W) and a nominal total specific impulse of 1390 seconds.

IV. Results/Discussion

As mentioned previously, the launch condition in EMTG was represented as departure from a circular orbit. The mission is concluded once the spacecraft rendezvous with the asteroid. Using the specific launch windows shown in Table 1, numerous iterations of EMTG scripts were run for each NEA, with launch windows linearly spread out by a year within each launch window. Overall mission lengths were set to an arbitrarily large number as length of the mission was not a constraint in this work. From each set of solutions for each NEA, priority was given to ones that spent less than or as close to 40 kg of propellant. From these, the solution with the shortest trip time was selected.

Each NEA-specific mission solution was tabulated for its propellant usage and total trip time. Figure 4 shows how much propellant each mission solution requires. The red line represents the 40 kg maximum allowable propellant usage. Figure 5 shows the total time each of the mission solutions require. Figure 6 in the Appendix shows the trajectories for all NEA mission solutions.

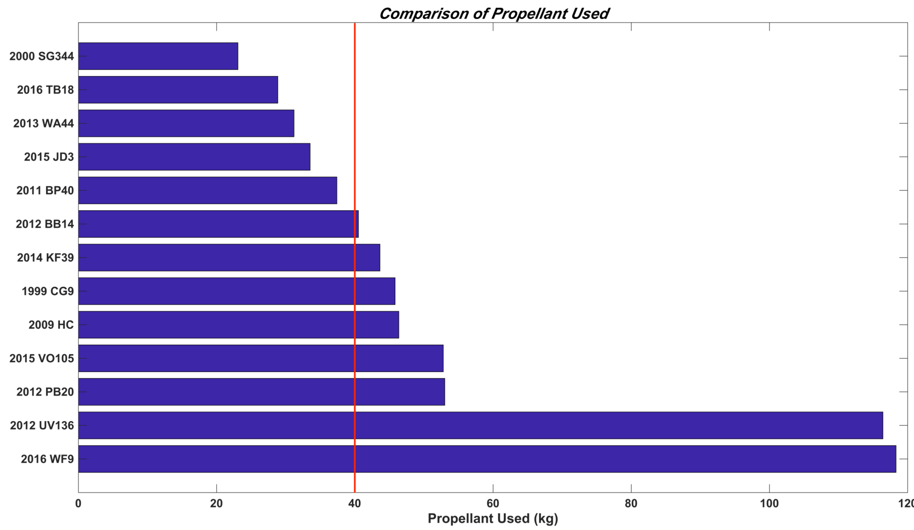

Figure 4: Propellant usage for each NEA.

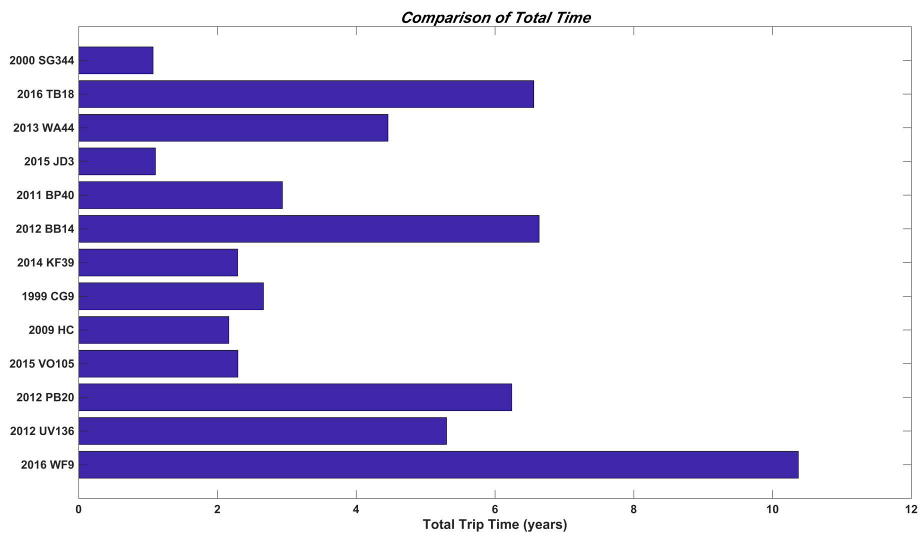

Figure 5: Total trip time for each NEA.

Out of the 13 mission solutions (one for each NEA) presented, 5 solutions (denoted with a border in Figure 6 located in Appendix A) used less than 40 kgs of propellant. Most of the solutions (with the exception of 2016 WF9) have a total trip within 7 years, with the minimum trip time being 1.1 years. Each of the 5 solutions uses a constant thrust from launch, resulting in a final orbit also very close to a circular orbit around the NEA of interest

3 out of the 5 asteroids (2016 TB18, 2000 SG344 and 2013 WA44) marked above have an average possible diameter around 50 meters. Asteroids 2011 BP40 and 2015 JD3 have a smaller average possible diameter around 39 meters. 2013 WA44 has the most uncertainty about its size (large bounds on diameter). SG344 was considered to have a high likelihood of impacting Earth and would create a 30 m wide impact crater assuming it doesn't explode in the atmosphere.[6]

The solutions can be compared to chemical propulsion missions to demonstrate the benefits of using a low power electric propulsion. Using the minimum ΔV's listed above, Table 2 shows the required propellant for taking a 180 kg wet payload to each of the NEAs. The table shows the propellant required for each NEA when using a 220 Isp mono-propellant hydrazine thruster and 310 Isp bi-propellant chemical propulsion system for referance. All results using a chemical propulsion system required much more than the 40 kg propellant limit set in the study.

NEA	Minimum Delta-V (km/s)	Approximate Propellant Required (220 Isp) (kg)	Approximate Propellant Required (310 Isp) (kg)
2016 TB18	5.55	165.6	150.0
2015 VO105	5.67	166.3	151.1
2015 JD3	4.84	160.1	142.2
2014 KF39	5.41	164.6	148.6
2013 WA44	5.96	168.0	153.7
2012 UV136	5.05	161.9	144.7
2012 PB20	5.28	163.7	147.2
2012 BB14	5.18	162.9	146.1
2011 BP40	5.84	167.3	152.6
2009 HC	4.43	156.0	136.9
2000 SG344	3.55	144.2	122.7
1999 CG9	5.25	163.4	146.9
2016 WF9	11.86	179.2	176.1

Table 2: Propellant required if using chemical propulsion

V. Conclusion

This study confirms the feasibility of an ESPA-class mission to reach NEAs with an electric propulsion system. A relatively simple commercial/off the shelf ESPA-class bus was used for the study, with a simple electric propulsion system based on a 200 watt BHT-200. This opens up the number of easily-reachable scientific targets for a category of missions that are critical to understanding the bodies that could pose catastrophic risks to our planet. The ESPA-class mission design significantly reduces the cost of these missions.

The constraints used in this study were limiting on all mission solutions by the ESPA-class mission design; with more propellant and/or a stronger thruster and/or a more efficient thruster, more NEAs would be accessible. There is a clear advantage to using electric propulsion instead of chemical propulsion when strict mission constraints on propellant and power are required. EMTG does provide an effective solution to rapidly determine the feasibility of this mission design, and can easily be used for new asteroids or for rapid turnaround.

A low-cost precursor mission studies, such as the missions described here, is significantly beneficial before preparing for more ambitious asteroid missions. Proper mapping, or even technology developments, such as grappling or gravity-tractoring, could also potentially be demonstrated with these mission classes.

Acknowledgments

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VI. Appendix A

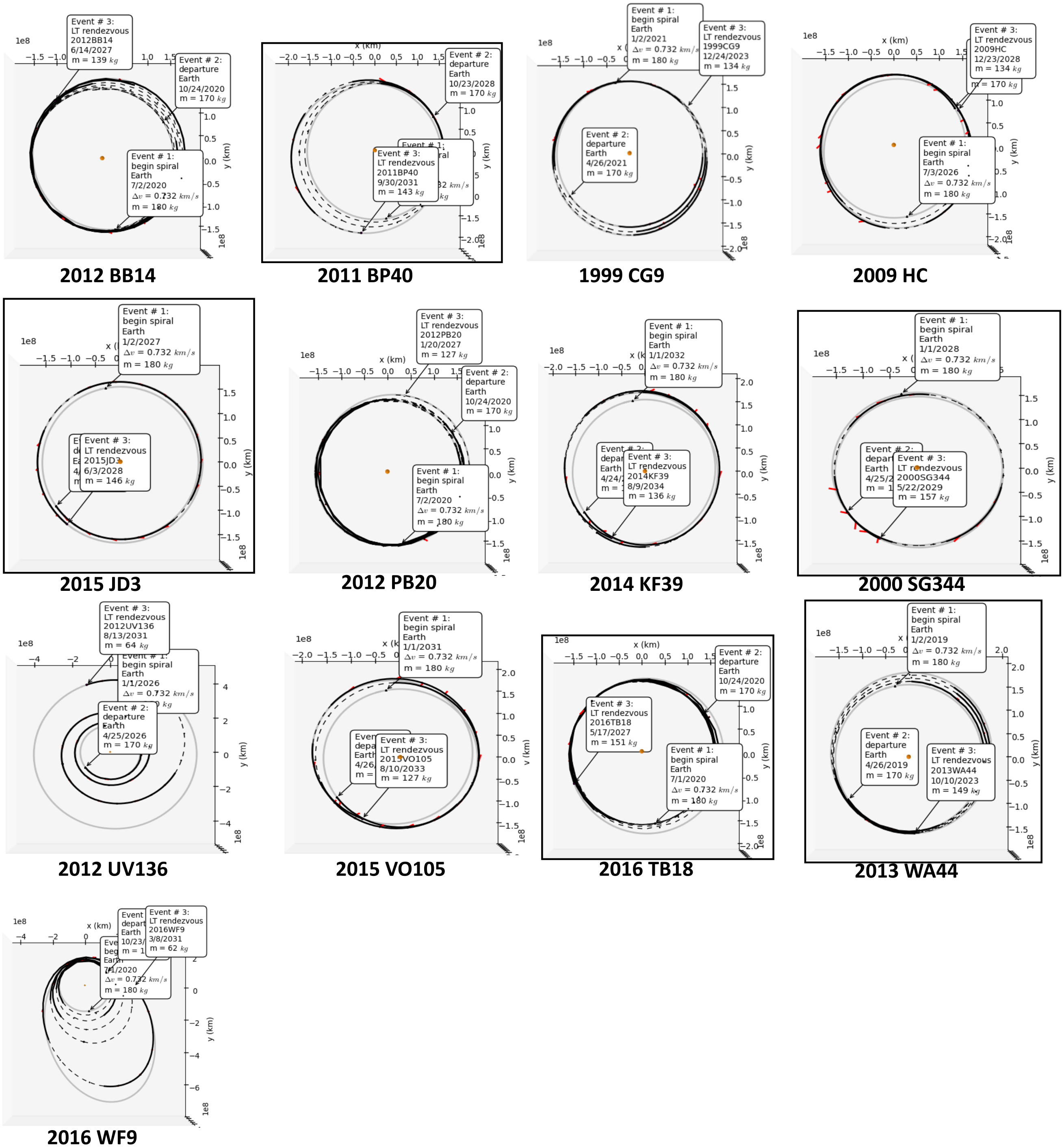

Figure 6: Mission Solutions for each NEA.

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[2] "Center for Near Earth Object Studies," Jet Propulsion Laboratory.

[3] Gates, M., "Asteroid Redirect Mission Update," NASA, July 2015.

[4] Englander, J. A., E. D. H. and Conway, B. A., "Global Optimization of Low-Thrust, Multiple-Flyby Trajectories at Medium and Medium-High Fidelity," .

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[6] "Sentry: Earth Impact Monitoring," Center for Near Earth Object Studies.